CHICXULUB EJECTA BLANKET DEPOSITS FROM BELIZE: THE ROLE OF VOLATILES IN LARGE IMPACTS.

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The Chicxulub impact into a thick sequence of carbonates and sulfates released over a trillion tons of volatiles. The importance of the explosive release of such a large mass of volatiles has been greatly underestimated in studies of ejects depositional processes. Proximal Chicxulub ejects blanket deposits recently discovered on Albion Island in Belize provide a key to understanding the role of volatile-rich target material during large impact events. The Belize ejecta comprise two stratigraphic units: (1) a spheroid bed composed mostly of dolomite spherules and altered glass shards and spherules in a dolomite and calcite silt matrix overlying Late Cretaceus dolomite; and (2) a dolomite silt matrix supported diamictite bed that contains dolomite clasts up to 8 m in diameter, many of which are polished and striated, altered glass shards, mudballs, and mud coated clasts. Mineralogic, isotopic, and comparative geological studies indicate that the spheroid bed was deposited by the vapor plume, and that the diamictite was deposited by combined ballistic and debris flow processes, Although the investigation continues, the l-db shortfall had to beaccomrnodated in Jupiter sequence planning.

All the earlier designed Jupiter approach sequences, JAA, JAB, JOE, were modified to accommodate the reduced telemetry performance. The early Orbital Tour sequences already completed will be modified during the planned Orbit Profile update period. Despite the l-db shortfall, these tests were successful in demonst rating the BVR operation at low spacecraft-sun separation angles (-6-10 degrees). The telecom experience with the Orbiter Deflection Maneuver in July 1995 resulted in changing the Relay/JOI telecom configuration to use a residual carrier signal to best maintain lock due to the extreme sensitivity of the link to thrust acceleration deviations.

The less-than-expected link performance is substantially mitigated by the previously negotiated addition of the Parkes antenna to the set of antennas supporting Galileo. Parkes is a 64-m radio telescope owned by the government of Australia and operated by the Commonwealth Scientific and Industrial Research Organization (CSIRO). It is located about 280 kilometers northwest of NASA's Canberra complex. Parkes has been previously used to support NASA'splanetary program, specifically the Voyager flybys of Uranus and Neptune, but the long-term commitment to Galileo represents a significant addition to Parkes' traditional role as a radio astronomy instrument and further recognizes the importance of Galileo's data to the international science community.

Parkes will support Galileo for each daily view period for 46 weeks over a one-year period, commencing with encounter C3 (1 November 1996) and running through orbit C1O (1 November 1997). To enable this support, NASA has funded CSIRO to modify the Parkes antenna to provide frequency agility—the capability to switch from radio astronomy to Galileo S-band and back immediately before and after each daily Galileo track. Thus, for the 46 weeks, Parkes will support both radio astronomy and Galileo on a daily basis. Parkes wilt be used exclusively for radio astronomy during the remaining six weeks; two weeks in January, 1997, when Galileo is in solar conjunction, and four weeks in the summer of 1997. When Parkes is arrayed with the Canberra antennas, it will increase the total data return for the day by over ten percent. Parkes can also serve as a backup tothe70-m station at Canberra, maintaining telemetry continuity (albeit at a lower data rate) if the 70-m antenna should become unavailable.

As part of the arrangement for use of Parkes, NASA is providing a modem S-band receiver for permanent installation at Parkes. At this writing, installation of this equipment is well underway and on schedule for first use at C3.

3. Phase 1 Inflight Computer Loading

The Inflight Load (IFL) of the Phase 1 Flight Software involved the complete reload of software in the two main computer control subsystems of the spacecraft (S/C), the Command and Data Subs ystem (CDS) and the Attitude and Articulation Control Subsystem (AACS). The IFL was accomplished over a period of 26 calendar days (including two adjacent days between the CDS and AACS loading . . . one for planned post-load clean up and one for a required thruster flush). The original IFL plan allocated a total of 43 calendar days for the IFL, but everything went so well that only one of 17 contingency days was used. The IFL was scheduled between two "cruise" sequences, Earth/Jupiter 8 (EJ-8) during which the Shoemaker-Levy 9 data playback was concluded, and EJ-9 during which the Probe Symbol Storage, part of the Phase 1 CDS functionality, would be validated inflight (see Figure 4). A detailed description of the Phase 1 flight software was presented in Reference 2 and will be summarized later. For the lFL period, the spacecraft was placed in the most quiescent state possible, with no flight sequence active and continuous tracking coverage provided. The IFL was scheduled to begin as soon as possible after the scheduled Phase 1 Flight Software system testing. This was the first time that the central spacecraft controlling computers of an interplanetary spacecraft were to be completely reloaded in flight. The IFL command packages had to be demonstrated to be perfect— the spacecraft hardware simulator on the ground, the testbed, was used for this purpose. It should be noted that the AACS memories were completely reloaded in Jaunary 1993 with the Phase OFlight Software. It was also important that the Phase 1 Flight software operate on the S/C for the longest time possible prior to the mission critical Probe Release and orbiter Deflection Maneuver scheduled for July 1995.

The initial IFL Conceptual Design Review was presented to the Project Office on May 3,1994. The AACS IFL command packages were approved for generation on November 3, 1994; and CDS IFL command packages on November 17,1994. Between the concept review and the command package generation approval, the subsystem experts expanded the concept into testable elements, and tested and modified these elements until they were convinced that they had an end-to-end product ready for formal command package generation. Following the generation of the command packages, system test began. AACS performed two separate tests; the first on December 6 and 7,1994, with a limited retest on February 14, 1995. CDS performed a single multishift system test on December 12–15, 1994. From concept approval to the completion of the Phase 1 IFL, nine months elapsed.

4.1 CDS Phase 1 Functionality

The CDS Phase 1 software changes provide: the capability to write edited Probe symbol and Doppler wind data to the CDS extended memory, new downlink telemetry rates at 8 bps and 16 bps, a new 7.68 kbps Probe data record format, improved Relay Readiness Configuration visibility in fixed format, engineering telemetry, 80-byte memory readouts (MROs) to decrease data return time, and more robust fault protection during Relay /JOI.

Fault protection changes include autonomous swapping of DMS tape recorder/CDS string connect ion in the event of a CDS string down during Probe Relay and modifications to the power bus Under Voltage recovery, AACS Power on Reset (1'01{) recovery, SAFING responses and thruster cluster thermal protection. Critical Mode SAFING response was also added, providing for the termination of the background sequence, if required, during operation of the Critical Engineering Sequence (CES).

The CDS Phase 1 Flight software was delivered on February 1, 1994. Subsequent to the delivery, a series of additional (RPM) propulsion subsystem and temperature control related fault protection changes were approved. They were all uplinked to the spacecraft separately. The CDS Phase 1 modified 100 of 510 software modules or approximately 20 percent; 24,764 of 91,144 bytes of code/data or approximately 27 pm-cent of all CDS addresses were changed.

The Phase 1 changes to the CDS and AACS were to be accomplished in a way that design redundancy would be maintained. For CDS the requirement for quad redundancy is met until Probe Relay symbols first start overwriting copies of the CDS code and data stored in the extended memory. At the end of Probe Relay the CDS is dual redundant (two full strings, two full normal memories both loaded with the I'base 1 flight software). Redundancy was important for two reasons. It provides the maximum robustness and greatly facilitated the loading of the software. Once the Phase 1 software was uplinked from the ground to the extended memory it could be copied to the normal memory, a more efficient process saving approximately 24 hours of commanding per CDS string. In the case of the AACS IFL, dual redundancy was maintained---a bus transfer process was used to load the B memory once the A memory had been uplinked from the ground and verified, saving approximately 60 hours in the overall AACS IFL process.

4.2 AACS Phase 1 Functionality

The AACS Phase 1 software included changes to compensate for star scanner radiation sensitivity including one star attitude determination and the use of the sun acquisition sensor as an attitude source[1], both for roll reference only during Relay. In addition, changes were made to expedite recovery from a possible Power-On Reset (POR) during Probe Relay by using the "saved" nominal spin rate in controlling the Spin Bearing Assembly (SBA) as soon as SBA data becomes available after an AACS POR to rapidly restore stator (RRA) pointing.

The Phase 1 software also includes "hooks" for the Phase 2 data compression software which will allow the uplink of Phase 2 software as a patch; a complete reload of AACS in Jupiter orbit is thus avoided. The delivery of the Phase 1 software occurred on June 1, 1994.

Subsequent to that delivery, two more changes were approved. The first involves 400-N burn termination should the accelerometers fail or data be lost due to an AACS POR. The second change involves the proper use of the pilot valve and latch valves for terminating a 400-N engine bum in the case of an MCS POR. These changes were delivered on December 5, 1994, and were included as patches to the IFL. Two further RPM related changes (patches) will be uplinked in October. One of these is to close the latch valves immediately after closing the pilot valve. The AACS Phase 1 modified 31 of 174 software modules or approximately 18 percent; 19,890 of 32,767 addresses or approximately 61 percent of all addresses were changed.

4.3 CDS IFL

In the simplest terms, the CDS IFL started with Phase O flight software loaded in all four CDS memories with CDS A- and B-strings running (both strings normally operate in parallel using the CDS A and CDS B normal memories). The memory test of the extended memories had been accomplished in August 1994 and was not repeated. Prior to loading the memories, each memory was, in turn, loaded with zeros.

The II--string extended memory was loaded with the Phase 1 software via **ground commands** and verified (Step 1). Variables from the B-string normal memory were copied to the B extended memory. The A-string extended memory was then loaded with the Phase 1 software via ground commands and verified. Variables from the A-string normal memory were copied to the A extended memory (Step 2). CDS A-string was taken down in preparation for swapping to the A-string extended memory. The B-string with the B normal memory with Phase O software was controlling the spacecraft.

The A-string normal memory was tested. Two hundred fift y-six bytes of zeros were uplinked to the CDS. Using 6MCOPY commands, the entire A-string normal memory was loaded with zeros. Then each byte was read with a set of loopers or groups of cyclic commands to test for memory read parity errors. The results were saved for later MRO. This process was repeated for all ones and later for a mix of ones and zeros. There were no memory read parity errors.

The Phase 1 software in A-string extended memory was then copied to the A-string normal memory and verified. The A-string with Phase 1 software on both the extended and normal memories was brought up, fault protection was reenabled, and the clocks were synched up with the B-string computer clock. The B-string was prime operating with Phase O and A-string was operating with Phase 1 software (Step 3).

Next the B-string was taken down in preparation for swapping from the B-string normal memory to the B-string extended memory. At this point the spacecraft was being controlled for the *first* time by the Phase 1 software; three of the four memories had been loaded with the Phase 1 software. The B-string normal memory was tested and then the Phase 1 software was copied from the B-string extended memory and verified (Step 4). The B-string was brought up, fault protection reenabled, the B-string clock was resynched with A-string clock. This completed the process,! Both A- and B-strings were operating with the normal memories with the Phase 1 software; Phase 1 software was also resident in both extended memories (see Figure 9).

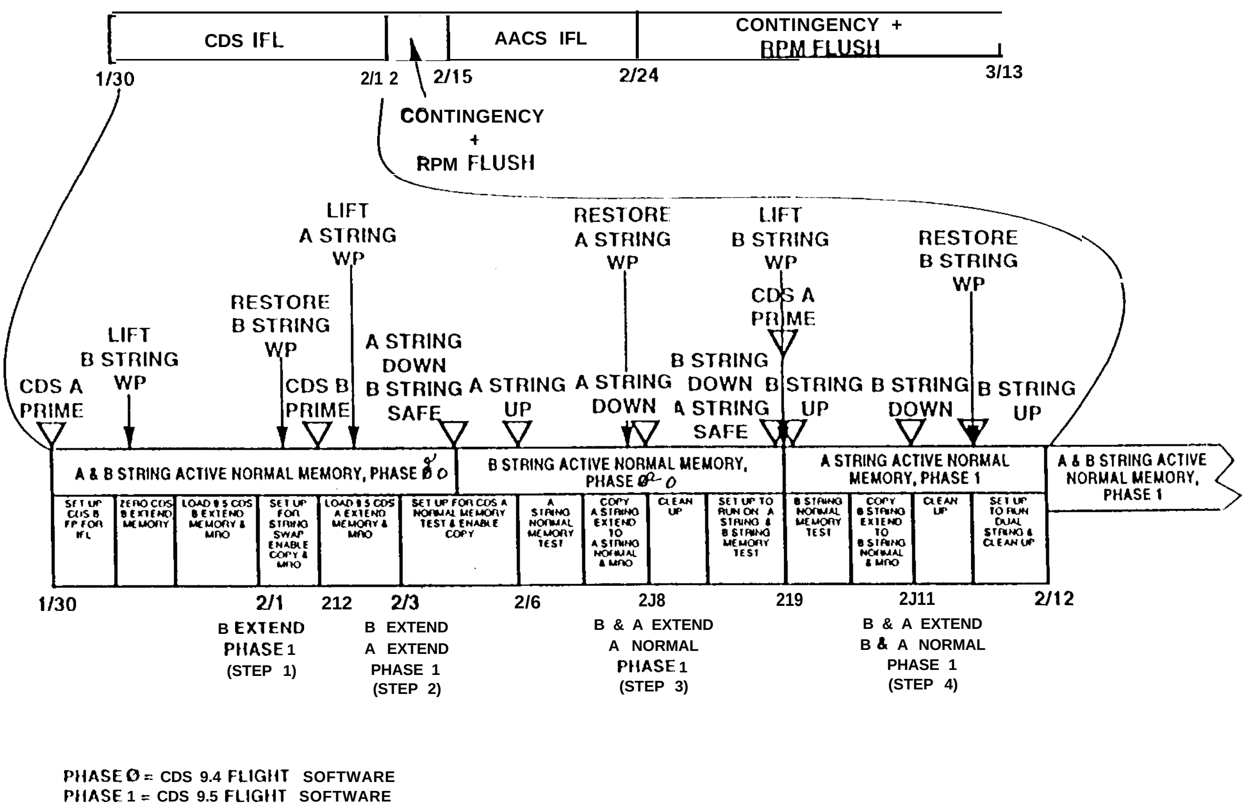
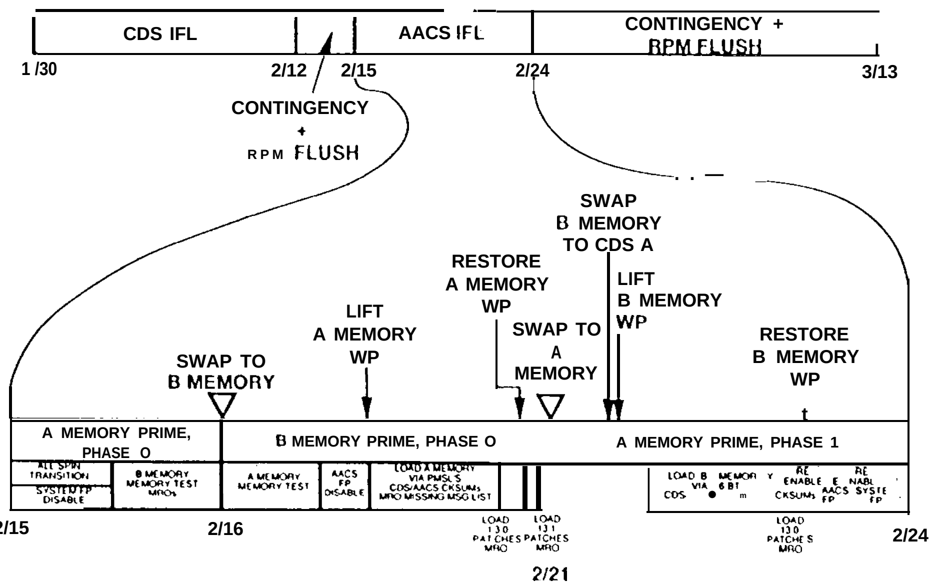


Figure 9. CDS *ϕ1 Inflight Load (IFL)*

The CDS IFL process was quite complex. A total of 31 command packages were generated and uplinked during the IFL. The CDS analysts were required to review frequent memory readouts and checksum telemetry on a very rigid schedule so that they could provide a Go recommendation at the required times, 27 Go/No-Go decisions were required—all were Go's.

4.4 AACS IFL

Loading of the AACS Phase 1 flight software was a little less complex than the CDS IFL process described in the preceding section. The process started with the Phase O Flight Software in both on-line and off-line memories. There are only two AACS memories to load, but this reduced complexity was offset partially by the decision to retest both memories before they were loaded, except for the B memory 1K non-addressable RAM. Note that both memories were previously tested prior to the Phase O IFL in 1993, and the off-line memory was tested again in August 1994. The spacecraft was set up in its least active mode (all-spin), and system fault protection was selectively disabled. The spacecraft attitude was controlled by the AACS A memory as the on-line memory, thus the B memory was inactive and could be tested. The memory test internally loads all ones, verifies the correct state; internally loads all zeros, verifies the correct state; and finally internally loads a known pattern of ones and zeros and again verifies the correct state. When a test is complete, the AACS memory test software automatically restores the original state of the memory, in this case, reloading Phase O flight software. These tests were done on small blocks of memory at a time. At this point, the A and B memories were switched, and the B memory became the on-line or active memory. Except for the test of non-addressable RAM, the memory test was repeated on the A memory. AACS fault protection was then disabled, and Phase 1 flight software was uplinked to the A memory and verified with check sums and selected memory readouts; a full memory readout was not practical since it would take too much time. Patches to the Phase 1 software, discussed previously, were uplinked and verified via MROs. A and B memories were again switched; the A memory became the on-line memory or active memory. At this time the spacecraft was first controlled by the Phase 1 AACS flight software. The B memory was loaded using data bus transfer commands or the functional equivalent of the CDSmemory copy. The CDS does a memory copy internally; the AACS equivalent depends on the CDS to transfer the memory load from the AACS A memory to the AACS B memory, via the spacecraft data bus. Fault protection was then enabled, completing the IFL process (Figure 10).



PHASE O = AACS 12.0 FLIGHT SOFTWARE
PHASE 1 = AACS 13.0 AND 13.1 FLIGHT SOFTWARE

Figure 10. AACS ϕ1 Inflight Load (IFL)

Fifteen separate command packages were generated and uplinked during the AACS IFL. The AACS analysts were required to confirm the correct status and provide a Go recommendation on a predetermined schedule. Had there been difficulties, the schedule would have been adjusted but everything went very well and 13 of 14 Go's were announced on time, as planned; ~~the fourteenth~~ one was delayed less than twenty-four hours until the next Canberra pass in order to reconcile an unexpected checksum word which turned out to be correct,

4.5 Summary

The IFL was completed much earlier than the Project anticipated. The few problems were managed in a very professional reamer and resolved to the satisfaction of the I'reject without requiring any significant delays, at any point, in the process. Heretofore, the largest command load uplinked to the spacecraft was the uplink of the Earth/Jupiter 6 (EJ-6) sequence, part A, which took 3 hours and 45 minutes. During the fourteen days of the CDS IFL, a total of 64 hours 39 minutes of commanding was required, averaging more than four and a half hours a day. During the ten days of the AACS IFL, a total of 25 hours and 11 minutes of commanding was required, averaging nearly two and a half hours a day. This is the equivalent of having sent the largest ever sequence to the spacecraft everyday, for *24* consecutive days. Records of this sort are not generally kept, but the Project knows of no other comparable level of commanding in the history of interplanetary y spacecraft operations, and it was done without error, and there was only one instance in which a command package had to be retransmitted.

5. Probe Operations

Probe operations this year focused on preparations for release. Two major tests were performed to characterize and predict Probe performance and determine its configuration for release. Although the Probe's internal command sequence could not be changed, there *were* options on the configuration of the Probe at release in the event of select anomalies. These options included disabling one of the two nearly redundant command and telemetry strings and/or disabling the Probe's coast timer. All of these options were contingency actions with serious science implications and would only have been taken in the event of serious, confirmed Probe faults which might otherwise have threatened the primary Probe mission.

One major consideration for the release configuration was the estimated Probe battery capacity. The Probe batteries consist of three lithium/sulfur dioxide (Li/SO_2) 13-cell modules which are non-rechargeable. The only measurement available from the flight batteries was the open circuit voltage (OCV) level of each of the three modules; there was no way to measure their capacity without discharging them. The open circuit voltage provides no measurement of the bat terycapacitybut does indicate if a cell has been shorted or opened.

To estimate the battery capacity, the Flight Descent Antecedent Test (FDAT) was developed early in the program. As battery lots were manufactured, they were stored at a variety of temperatures for lifetime testing. Cells and modules have been discharged regularly to determine the long-term degradation as a function of time and temperature, and have shown only a small variation in capacity from module to module within each lot even after the long storage times associated with the mission flight time. The flight lot, Lot 10, was built just before launch in 1989. Several of its flight spare modules and cells have been stored at the same temperature profile that the Probe flight batteries experienced since launch, including the temperature excursions which occurred during the High-Gain Antenna (HGA) cooling and warming turns.

For the FDAT, three flight-lot battery modules were put under a real-time simulated load and temperature profile. The 155-day coast timer load began in September 1994, with the batteries maintained at 0°C. The simulation of the pre-entry, entry, and descent profile was made on February 21, 1995. Results showed the ground flight-spare batteries had a usable capacity of 19.3 A·hr. Given that the flight batteries would be 10 months older at Jupiter encounter, the flight batteries are estimated to have a usable capacity of 19.2 A·hr, sufficient for 76 minutes of descent operations. All data from the long-term ground battery testing showed that the batteries are very consistent from module to module and the worst-case estimated variations in performance due to variations in initial battery capacity were conservatively estimated to be ±12 minutes (double the largest variation seen in the ground data).

The final OCV measurements before release were collected in March 1995 during an Abbreviated Systems Functional Test (ASFT). The ASFT was designed in 1991-92 to check out a limited set of Probe functions at the reduced data rate of the Orbiter low-gain antenna. Its purpose was to collect only the data required to properly configure the Probe for release. This included the three OCV measurements as well as calibration data from the Atmosphere Structure Instrument (ASI). The ASI data was used to determine if a change in the prime ASI accelerometer was required before release. The procedure also pumped down the Neutral Mass Spectrometer ionization chamber, although without telemetry verification. Lastly, the ASFT provided a final end-to-end test of the orbiter FSW upgrade which enables the storage of the highest priority Probe data in the CDS extended memories (Probe symbol storage) as a backup to the Probe data storage on the tape recorder during Relay.

The ASFT was executed on March 16,1995. All Probe data was nominal. The OCV measurements were "spot-on"; clearly no battery cell failures. The ASI accelerometer data indicated normal operation so no reconfiguration was required. The edited symbol data was compared to the unedited tape recorded data and demonstrated' the efficiency and acceptability of this data storage technique. The symbol data storage capability was a major portion of the Phase 1 software upgrade uplinked in February, and allows for Probe data bits (referred to as symbols, since they are encoded bits, and not raw data) to be efficiently stored in the CDS memory. Neither receiver data nor filler bits are stored, reducing the data rate from the 6.48 kbps from the Relay Radio Hardware (RRH) to the 512 bps (256/string) data rate from the Probe. A symbol-to-symbol comparison between the two data paths showed an exact match, except for two short intervals of data corruption which were determined to be due to low signal margin in the spacecraft-to-ground link during the tape recorder readout.

As part of an overall review of the Probe Release geometry and dynamics, revisiting some immediately pre-launch incomplete mass properties documentation suggested that the Probe wobble and nutation upon release might be significantly above expectations—and could exceed the established requirements for the entry angle of attack. A thorough investigation of this finding was performed. Although it is believed that this "final" documentation is in error, irrefutable evidence of this error could not be found and the increased wobble and nutation was conservatively treated as real. Probe entry dynamics analyses were reviewed by Project Galileo engineers and by Ames and Langley aerodynamists. It was determined that although the established requirements might be exceeded, there existed substantial margin in the requirements, and no threat to the Probe existed from potentially exceeding wobble and nutation. The specification for maximum error in entry angle of attack was increased from 6.0° to 7.3°.

Probe Release activities began on July 1, 1995 when the Orbiter RRH oscillators were powered on. The Probe Release sequence commanding began on July 5 when the Probe Power-Up sequence was executed, turning on the Probe receivers and the Probe Data and Command Processor. Although no data was collected from the Probe during the Power Up, Probe temperatures and current data from the Orbiter confirmed that all commands executed as expected. At the end of Power Up, the B-string receiver and oscillator were powered off, as the release sequence used A-string commanding only. The coast timer was loaded and started, with a six-minute snap of Probe symbols collected into the CDS, starting just before the coast timer start. During a sequence hold, the Enable Hold, the data from the CDS was downlinked twice, to verify the coast timer load and first decrement. The first downlink was processed nominally and showed perfect performance of the Probe and coast timer.

The Go command was transmitted as scheduled and the second portion of the release sequence was executed on July 7, when the Probe's coast and G-switch power buses were connected to the internal batteries. Six more minutes of Probe symbol data were collected by the CDS, and the Probe was then turned off, except for the coast timer which will time-out just before Jupiter encounter, initiating the Probe's internal, pre-stored pre-entry science command sequence. The sequence then entered its second hold, the Cable-Cut hold, while data was reviewed to verif y Probe state before cutting the Probe umbilical cable. Proper configuration of the Probe coast timer and G-switch relays was verified, and Orbiter current and Probe temperatures were used to verify proper power-down of the Probe. The CDS data was downlinked during the hold and showed continued normal operation of the Probe, with the coast timer continuing to count down nominally.

The second Go command was transmitted as scheduled and the third part of the Release sequence began on July 10. The A-string RRH receiver and oscillators were powered off, and then the umbilical cable between the Orbiter and the Probe was cut at July 10, 10:26 P.M. Pacific Daylight Time ERT. Verification of the successful cut was made when all telemetry lines from the Probe (three temperature and eight relay states) indicated an open circuit.

The Orbiter then began reconfiguring for Probe Release by executing a Spacecraft Inertial Turn (SITURN) to the Release attitude and spinning up to the nominal release spin rate of 10.5 rpm. After these events were confirmed, the last Go command was transmitted as scheduled early on July 12. The Probe was released on schedule July 12 at 11:06 P.M. Pacific Daylight Time ERT. Release was immediately confirmed by Doppler data showing the velocity change of the Orbiter. Telemetry verification followed shortly after **when two mechanical** separation switches indicated that the Probe had been released. Initial analysis of the Doppler data showed a separation velocity only 3 percent less than expected, well within the 6 percent tolerance. Although detailed data analysis is not yet complete, the Probe trajectory and attitude are confirmed to be well within margins for a successful entry on December 7th.

6. Probe Navigation

Navigation has been excellent throughout the flight. This year marked the most critical of all navigation operations—Probe Release. The Probe has no control system and is not commendable after release; therefore, the Probe entry trajectory had to be established by the Orbiter before release. Accordingly, the release impulse and all gravitational and non-gravitational forces acting on the Probe following release had to be precisely model led in the pre-release targeting of the Probe. As seen in Figure 4, TCM-23 and TCM-24 were scheduled to refine the spacecraft trajectory before release. The Orbit Determination based on the DSN S-band tracking and the TCM-23 execution were so accurate that the final pm-release TCM (-24) was cancelled.

The critical parameters for Probe entry are the entry flight path angle (FPA), angle-of-attack (AOA), and entry time. FPA and AOA are critical for surviving entry—loads and heating. Jupiter is by far the most difficult body in the solar system to enter directly because its tremendous gravity accelerates any arriving object to over 180,000 km / h-five times faster than any previous planetary entries including Earth returns. The FPA corridor is -8.6 ±1.4 deg (99 percent); AOA specification is to not exceed 7.3 deg (99 percent). The Orbiter also had to control AOA by turning to align the centerline / spin axis to the inertial orientation of what will be the atmosphere-relative-entry-velocity vector, and then spinning up the entire spacecraft to 10.5 rpm prior to release. The high spin is needed to adequately maintain the Probe attitude during its five-month solo flight to Jupiter. A feature of this process was slightly adjusting the two Radioisotope Thermoelectric Generator (RTG) booms to rebalance the spacecraft so that the spin principal axis-of-inertiaat 10.5 rpm would realigned with the geometrical centerline. This minimized separation disturbances to protect clearances and limit AOA errors.

Spin-up is done by pulsing a single 1O-N roll thruster at a 1.3 / 3.9-s duty cycle. This results in a small residual lateral velocity error (the AV vectors for the many pulses do not perfectly close) that is ironically the largest a priori contributor to the FPA error. Next largest is the uncertainty in the spring separation impulse. The Probe Release was flawless. Tables la and lb show the critical parameters following both TCM-23 and Release based on DSN Orbiter tracking and telemetry. All values are well within specification including current uncertainty estimates. The Probe is on a near-perfect trajectory and is properly oriented for entry.

Figure 7 illustrates the Probe and Orbiter arrival geometry. The Orbiter must be in the proper overflight position to receive the Probe Relay Link and later insert itself into orbit. Key performance constraints on the Orbiter trajectory are that perijove

Table la. Probe Delivery Status

Parameter	Target	Miss After TCM-23 With Nominal Release Modeled	CBE Including Estimated Release Errors		
			Miss	Knowledge +/- 99%	Requirement +/- 99%
Entry Flight Path Angle, deg	-8.60	+.01	+0.27	0.15	1.4
Latitude (JMED), deg	6.57	<,01	-0.02	0.03	0.5
Entry Time (UTC), 7-DEC-95	22:04:26	+2 sec	-11 sec	98 sec	480 sec
ENTRY ANGLE-OF-ATTACK ERROR: PATH ERROR CONTRIBUTION IS WELL WITHIN THE 1.0 DEG OF ERROR BUDGET ALLOCATED TO NAVIGATION (PATH ACCURACY)					

Table 1 b. Orbiter Control of Release Attitude

. POINTING ERROR= 0.16 DEG • WOBBLE (Y-AXIS)= 0.0S DEG	vs. 99% ANGLE-OF-ATTACK SPEC = 7.3 DEG"
• SPIN RATE= 10.S0 RPM vs. 10.5 ±0.5 RPM (99%)	
' A-O-A ERROR IS DOMINATED BY PROBE PARAMETERS (PROBE NUTATION, PROBE WOBBLE, AND PROBE MAGNETIC IMBALANCE)	

shall be 4.0 R_J and a 1,000 km altitude Io flyby shall be used to 'obtain a "gravity-assist"—both to minimize the orbit insertion AV required. The Orbiter radiation budget precludes perijove below 4.0 R_J. These two constraints determine the exact time of Io flyby and therefore the exact time the Orbiter wiLl be at the relay start position (for a given Relay Link geometrical design). Remarkably then, it is the Orbiter alone that determines when the Relay must start and, therefore, dictates the Probe entry time! Note in Table la that entry time was easily controlled within the relatively large allowable tolerance.

The current best estimate of AOA control is 6.5 deg (99 percent).

7. First Use of the 400-N Main Engine and the Orbiter Deflection Maneuver

As described earlier, the Orbiter had to establish the Probe entry trajectory. Thus, following Probe Release the Orbiter was also on a Jupiter entry trajectory. The Orbiter Deflection Maneuver (ODM) was required to "re-target" the Orbiter to its Io flyby aim point to establish the required trajectory for Probe Relay and JOI. From Project inception it has been the mission plan to use the Orbiter 400-N main engine for the first time to perform the ODM. It was impossible to use the engine earlier because the Probe was mounted right in front of the 400-N nozzle. Furthermore, there was a constraint that the wetted lifetime of the engine not exceed nine months, which is consistent with the three intended 400-N maneuvers (the three largest of the mission)—ODM at five months before JOI, JOI, and Perijove Raise (PJR) three months after JOI. Orginally, the basis for using the 400-N engine for ODM was the 10 percent higher Isp compared to the 10-N thrusters. The ODM could have been performed with the 1O-N thrusters for a penalty of ~5kg of propellant. In actuality, since the Probe data cannot be returned until after orbit insertion due to the high-gain antenna failure, inflight "qualification" and characterization of the 400-N engine at the ODM became vastly more important than the propellant savings. The JOI is the only maneuver that absolutely requires the 400-N engine. Originally, with the Probe data returned before JOI there was little to lose by attempting the JOI, regardless of the health of the engine. Now, it was essential to do everything reasonable to demonstrate proper engine operation, if at all possible, prior to committing to the JOI burn.

The Galileo propulsion system was designed and built by MBB (now Daimler–Benz Aerospace) under contract to the German Space Agency (DARA) that supplied it toNASA free of charge as Germany's largest contribution to the joint U.S.–FRG Galileo Project. It is called the RPM for Retro Propulsion Module, which is a misnomer since it provides all attitude control torquing and all AV impulse. It"is a helium-pressure-fed bi-propellant (MMH andNTO) system. The two fuel and two oxidizer tanks are common to all twelve IO-N thrusters and to the 400-N engine. They are also common to the pressurization system. Three pairs of solenoid-operated latch valves isolate the propellant supply–one pair for half the thrusters on an A-branch, one pair for the redundant half on a B-branch, and a separate pair for the 400-N engine. The flawless and very extensive operation of the 1O-N thrusters throughout the flight had already partially demonstrated some aspects of the 400-N operation, but most of it, including the engine itself, had yet to be operated in flight. The Project in concert with DARA and DASA worked intensively, particularly this past year, preparing for the first 400-N firing inflight, and the subsequent JOI.

An exhaustive analysis was done for all elements of 400-N engine operation, including failure modes and workaround contingencies. Ultimately, the Project established a very elaborate decision tree that specified the best course of action for every plausible failure. Some failure scenarios would have deferred the first 400-N burn to JOI with extraordinary valve operations, and some dictated that the JOI would have to be abandoned (i.e., loss of Orbiter mission). All the work and the ultimate strategy was reviewed and endorsed by a NASA-selected independent review board.

The 400-N engine propellant valves are pneumatically actuated by the same helium pressurant supply used for the propellant tanks. An electrically operated hi-stable pilot valve opens to flow helium to the engine valves, and closes to vent the valve pressurant overboard. The helium was isolated from the pilot valve by a pyro valve until first use. If the pilot valve leaked when pressurized, all the helium could be lost overboard and the propellant pressure feed would be lost. The ODM and TCM'S to get to and through Relay could be done in blowdown, but JOI and the Orbiter mission would be lost. The RPM design incorporated a pair of pyro valves that were to be fired to cap the pilot valve vent port and flow the helium through the pilot valve backward to open the engine valvesin the event the pyro isolation valve didn't open. Recognizing the catastrophic nature of a pilot valve leak, this feature was used to prevent it. The spacecraft autonomous fault protection was augmented with a patch that would fire the contingency valve pair, capping the vent if the helium supply pressure (i.e., above the regulator) dropped below a pre-set threshold, This would save the mission, but had the undesirable consequence of permanently opening the 400-N engine valves. Then the engine could only be started and stopped with the latch valves. Extensive 400-N firing tests were performed in Germany this past year with a flight spare engine to demonstrate the safety of latch valve operation.

In the above contingency---open engine valves— a latch valve leak would be catastrophic. The RPM design also provided pyro valves that could be opened to marry the 400-N propellant lines to the A-branch thruster lines in the event the 400-N latch valves wouldn't open. But if the engine valves were already permanently open, this bypass would preclude any further use of A-branch thrusters, since the 4(K)N would fire any time the A-branch latch valves were open. Accordingly, the strategy was to demonstrate the pilot valve was leak tight before marrying the propellant branches would be considered.

While the identical lot latch valves onthe thruster branches had worked flawlessly hundreds of times, the 400-N latch valves had only been cycled once to vent the propellant lines during the launch sequence. The first action in the ODM sequence was to cycle the 400-N" latch valves 25 times to demonstrate their reliable operation. Had they been faulty, only if there was no open threat, would pilot valve pressurization proceed, and then if the pilot valve was leak tight, the bypass would be fired to proceed with the 400-N ODM on the A-branch. Otherwise, the first 4(X)-N bum would be deferred to JOI. Happily, no such contingencies occurred.

Following the inflight demonstration of the 400-N latch valves and the successful pressurization of the pilot valve, it was necessary to do a 2-second "wake-up" (WUB) test bum of the engine. This was long enough to obtain data on proper combustion (i.e., both propellants reached the chamber and burned properly) and short enough that no damage could occur to the engine or spacecraft. Particularly, it was necessary to ensure that both propellants were flowing into the chamber before committing to the 5-minute ODM burn. Aborts could not be commanded from the ground since the roundtrip signal time was 76 minutes.

The 2-second WUB was completely satisfactory. It was suprising at first that it was about 13 percent under predict. Subsequent analysis and the very near nominat ODM main bum strongly suggest the underperformance was due to the long- term helium permeation through the latch valves so that there was helium bubble froth in the propellant lines that was consumed in the first 1.5 seconds that "cleared" the line volume. The lines were purposely not vented in order to avoid spacecraft contamination and minimize water hammer.

Finally, before committing to the 5-minute bum, it was necessary to verify that helium was flowing through the fuel checkvalve. There is a checkvalve in the pressurant line to the fuel tanks and one in the line to the oxidizer tanks to limit upstream migration of propellant vapors and their mingling. There was a remote concern that the reverse pressure spike seen by the checkvalves at pilot valve pressurization could jam a checkvalve closed. Furthermore, the pressurization system had been in lockup for nearly two years because so little propellant had been used in that time. Accordingly, it was a prerequisite to demonstrate pressure increase in the fuel tanks *after* pilot valve pressurization. This would demonstrate both the fuel checkvalve and the pressurant regulator flow. If the fuel checkvalve stuck closed, it would not be safe to bum for five minutes because fuel pressure would go dangerously far out of the engine operating box. If the regulator was stuck, it would be desirable to do part of theODM on the spare regulator to demonstrate it. Thus, the ODM was to be divided into two roughly equal segments days apart if the fuel pressure didn't rise. The post-Probe release spin-down was delayed until after pilot valve pressurization so that it would add to the attitude maneuvers and ODM spin-up propellant expulsion which in conjunction with propellant heaters management would maximize seeing the system "crack" after pilot valve pressurization. The cracking was seen immediately after pressurization so this criterion was satisfied at the earliest possible moment, although it wasn't required until just before the 5-minute bum commitment. Incidentally, the entire ODM could be safely done in blowdown on the *oxidizer* side; however, if either checkvalve never opened, then JOI could not be attempted.

Table 2. Propellant Margin calculation
(Post-ODM Epoch: 8/1/95)

Event	I_{sp} (s)	ΔV (m/s)	M_i (kg)	M_f (kg)	ΔM (kg)
Completed Events:					
Drop Adapter			2717.2	2561.2	156.0
Drop Instrument Covers			2561.2	2560.4	0.8
ΔV Propellant (10 N)		131.0	2560.4	2433.7	126.7
Attitude & Spin Control			2433.7	2406.5	27.1
HGA Anomaly Activities			2406.5	2355.3	51.3
RPM Line Flushing			2355.3	2349.7	5.6
Probe Release			2349.7	2010.8	338.9
ODM	308.5	61.5	2010.8	1969.8	40.9
Future Events:					
I/P Statistical ΔV	270.7	4.5	1969.8	1966.5	3.3
I/P Deterministic ΔV	270.7	0.0	1966.5	1966.5	0.0
I/P Attitude & Spin Control			1966.5	1966.2	0.3
I/P RPM Line Flushing			1966.2	1966.0	0.2
JOI	308.5	643.8	1966.0	1589.2	376.9
OTM-1 + OTM-2	270.7	3.0	1589.2	1587.4	1.8
PJR	308.5	375.4	1587.4	1402.1	185.2
Tour ΔV	270.7	67.3	1402.1	1367.0	35.1
Tour Attitude & Spin Control			1367.0	1352.1	14.9
Tour RPM Line Flushing			1352.1	1350.3	1.8
Science Turns			1350.3	1330.3	20.0
Project Manager Reserves			1330.3	1317.3	13.0
		End-of-Mission Mass:	1317.3		
		Orbiter "Burnout" Mass:	–1 296.5		
		Propellant Margin:	20.9		

Notes: I_{sp} range for completed ΔVs: 269.4 -274.4 s.
Values may not add because of rounding.

M_i = spacecraft mass before event.
M_f = spacecraft mass after event.
ΔM = spacecraft mass change,

10. Relay/JOI Critical Engineering Sequence Salient Updates

The Relay/JOI Critical Engineering Sequence (CES) and integrated autonomous fault protection strategy /implementation were changed significantly to further increase overall sequence execution reliability/robustness. This activity dealt with protection against very unlikely faults/faults occurring in a small vulnerability window; the protection against the more likely faults has been in the CES design for years. Most of the updates resulted from improved understanding of potential RPM (propulsion subsystem)-related failure modes/ consequences and incorporation of fixes required to resolve problems revealed in system testing of the CES. The following paragraphs briefly describe the significant changes made this past year.

10.1 AACS Updates for Probe Relay

Because capturing the Probe relay data requires reliable and accurate Orbiter stator (despun section) control for pointing the relay antenna, changes were made to disable the "inertial observer" portion of attitude control fault protection as part of Relay Readiness Configuration (RRC) activity. This gyro fault protection protects against extremely low likelihood faults. It uses star scanner output to detect excessive gyro drift. The other twelve gyro-related fault protection algorithms remain enabled. With the inertial observer disabled, the attitude control subsystem stays on gyro-based control rather than switching to star scanner information should a pointing discrepancy be detected between star scanner and gyro-based data. This change was made based on an increased concern for the proper operation of the star scanner deep in the Jupiter radiation environment during Probe relay. Given the excellent performance of the gyros throughout the mission, the prospect of excessive gyro drift in the radiation environment is far less than radiation induced fallacious star scanner output. However, if any other gyro fault monitor trips, control will switch to the star scanner output and, if that is faulty, ultimately, to the sun acquisition sensor for roll reference'.

10.2 RPM 400-N Latch Valves Open Time

The 400-N Latch Valves (LVs) will be commanded open at JOI bum start –1 minute instead of the earlier opening time at JOI-3 days. The earlier open time was previously selected to permit ground recovery time to use the backup 1O-N LVS (to feed the 400-N engine) if the 400-N LVS failed to open. Recent inflight 400-N LV tests and the ODM in July 1995 provide high confidence that the LVS witl open when commanded. Opening the LVS at JOI -1 minute protects the collection of Probe Relay and Io data should the pilot valve inadvertently open. Although such a fault is considered extremely unlikely, the total mission would be lost with the previous LV opening approach if the pilot valve inadvertently opened anytime during the 3-day period when the LVS were open.

10,3 RPM Overpressure JOI Update

Changes to RPM-related fault protection include the re-enabling of the RPM overpressure algorithm near the end of the JOI bum to particularly protect against the possibility of the RPM pressure regulator sticking open in the 400-N bum pressurant flow position during the long JOI burn, and then—after engine stop—overpressurizing the propellant tanks. If not protected, the tanks could reach a pressure of about 22 bar (burst disk limit). The consequences of this would be loss of helium and a threat to 1O-N thruster health/safety when subsequent required thruster operation would be at anomalously high pressure. Currently, the fault protection threshold limit is safely set at 19 bar but can be updated by ground command, if necessary.

Previously, post-JOI overpressure protection was precluded by a constraint not to fire a pressurant pyro isolation valve in the high radiation environment. To handle more critical contingencies, an investigation this past year demonstrated that the pyres can be safely fired in high radiation, thus enabling this added protection as well.

10.4 RPM Helium Loss JOI Update

Another RPM-related fault protection change involves enabling the "helium loss" algorithm during the initial portion (1 to 17 minutes) of the 49-minute JOI bum to protect against a large loss of helium pressurant that would result in engine shutdown (loss of servo pressure) / inadequate propellant feed pressurization before orbit capture. The helium tank pressure threshold limit will be set at 80-bar prior to the JOI bum. The 80 bar trip limit to 17 minutes into the bum also preserves enough helium to feed the remaining propellant during the orbital mission. After the initial 17 minutes, the helium loss atgorithm is autonomously disabled by the CES because of the large nominal pressure drop over the JOI bum. Without this protection, a leaking pilot valve in either the open or closed position (B port seat or A port seat) vents helium overboard. The protection response caps the vent and unfortunately causes the engine valves to be open permanently; thereafter, the 400-N engine must be controlled with its latch valves. Shortly after JOI bum completion, ground commands previously time-buffered in the CDS will re-enable the algorithm at a lower limit to protect against possible fast loss of helium with the pilot valve back in the non-bum position. The post-JOI helium loss threshold limit is TBD.

10.5 RPM Continuously Powered Valve Update

In addition to the aforementioned changes, the continuously powered latch valve fault protection algorithm is now enabled throughout JOI. The earlier strategy disabled the Continuously Powered Valve (CPV) fault protection algorithm from the start of spin-up through the JOI bum and then re-enabled it shortly after completion of the JOI bum. Upon revisiting the earlier strategy, it was determined that should a CPV fault occur, the most reliable strategy is to have the algorithm enabled so that electronic drive power will be autonomously removed from the valve, precluding overheating which could lead to the total loss of the spacecraft. Unfortunately, with CPV enabled, should an untimely drive electronics fault occur, the fault response turns off the drive electronics and precludes opening the 400-N LVS, resulting in loss of JOI. As bad as this would be, the updated approach preserves the Probe Relay and Io data which is stored on the tape recorder for later return to Earth but leaves the JOI vulnerable to an extremely unlikely failure.

An important conclusion of the contingency planning is that the latest opportunity to begin an attempt to recover a down CDS string is 72 hours before Relay. Furthermore, an attempt so close to Relay is viable only if the string is down due to the well understood (CDS) TBR fault. String recovery would be in order to restore the dual string execution of the Relay/JOI CES. Within 72 hours of Relay it is more reliable to depend on the single string execution of the CES than to attempt a recovery. Another TBR will not bring down the surviving single string. Fortuitously, this late recovery allows reloading the arrival science observation sequence that is canceled when a string goes down.

12. Ida-Like Anomaly Protection

During the Ida encounter in 1993, about five hours before closest approach, the gyros tripped off due to a high-rate fault. The spacecraft properly switched autonomously to star-scanner and actuator encoders based instrument pointing, which was completely successful. No hint of this anomaly, which was discussed in both References 1 and 2, has ever occurred since, in spite of efforts to repeat it for diagnostics. This anomaly had grave implications for Jupiter approach and arrival. Six times on approach to Ida, max torque was anomalously applied briefly (-1 second) to both the clock and cone scan platform actuators. This causes a 32-watt power demand increase; the usual flight margin of 20-watts would be greatly exceeded, causing power bus undervoltage trip which would autonomously terminate the observation sequence. The Critical Engineering Sequence (CES) that performs Probe Relay and JOI would continue, but all the remaining approach science observations would be lost.

With great difficulty, the approach phase power management has been reworked to provide enough margin that an additional anomalous 32-watt demand will not cause an undervoltage, so the observation sequence would continue albeit with a brief pointing disturbance. Furthermore, everything possible has been done to avoid the anomaly—given our incomplete understanding of what actually caused it.

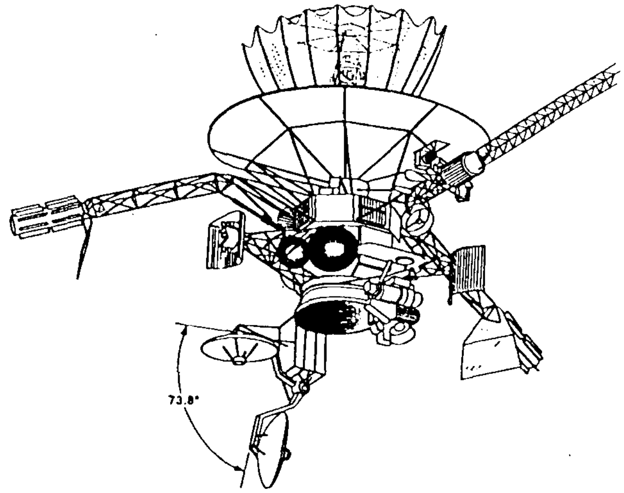
There are two leading potential causes. The first is very pathological. A bit error in the Most Significant Bit (MSB) of the cone encoder data word "tells" the Flight Software that the scan platform is beyond 180 deg by the amount it is actually short of 180 deg. When the platform is at less than 153-deg cone the logic checks if the "measured" cone is at or past the cone stop at 210 deg. If so, the cone control 100p switches from gyros to encoder and calculates rate error using encoder measurements only. The fallacious reflection across 180 deg causes a huge rate error and consequent max cone torque command. This trips gyros off and causes the platform clock to switch to encoder control. When the average of the desired and measured cone angle is 180 deg—which can't happen in reality, but the reflection results in exactly that-a divide by zero occurs causing a huge, fallacious clock-angle-indicated error resulting in max clock actuator torque. To protect the preferred, inertial pointing mode for Relay against this scenario, following the Io encounter the platform will be parked at O deg cone and the inertial (gyros) mode will be recommended by the CES just in case the gyro trip had previously occurred.

A bit flip in the onboard Target Motion Compensation (TMC) calculations for platform pointing could also cause the symptoms. Accordingly, the sequence has been carefully checked to ensure that TMC is not used after Io.

With these safeguards, the arrival Europa and Io observations would be safely performed in the presence of this Ida-like scan platform anomal y. The previously planned Europa and Io demonstration slewing on the flight spacecraft has been cancelled because even if the anomaly was demonstrated—which is extremely unlikely-we would still perform the observations now that we have adequate safeguards in place. The sequences have already passed all the ground checks.

13. Relay Radio Antenna Positioning

As illustrated in Figure 11, the Relay Radio Antenna (RRA) mounted on the Orbiter to receive the Probe signal was rotated to its initial position in August. The cone angle of the RRA-the angle between its boresight and the Orbiter +z axis-is controlled by redundant drive motors and potentiometers at the hinge point. Verification of proper pointing is based on agreement between the calibrated motor ruining speeds and times and the potentiometer values. Slew times are also constrained to protect against potentiometer faults. The RRA stays at its initial position for the first 32-minutes of the Relay, and then it is stepped four times in 2.5 to 3.5 deg steps over the remaining 43 minutes of the Link. The stepping profile was used three times in combination with other slews to get to the initial position inamannerthat further verified the stepping performance. The inflight slewing was first tested in a 1993 special test. The RRA will not be moved again until 32 minutes into the Relay Link.



NOTE: RRA WAS PURPOSELY LEFT -7° OUT OF STOW (0°) AFTER 1993 SLEW TEST. INITIAL POSITIONING SLEWS PUT BORESIGHT 73.8 OFF S/C + Z.

Figure 11. Relay *Radio Antenna* Positioning

14. One Star Attitude Determination (OSAD) Flight Test

As described in Reference 1, new concerns regarding the reliable operation of the Star Scanner in the intense Jupiter radiation environment during Probe Relay resulted in modification of the AACS FSW. The new “one-star” capability will use only the most detectable star, Canopus, to provide the most robust possible Star-Scanner-based roll reference. Nominally, the roll reference will be the gyros, but gyro faults and some other faults require a backup reference. Since AACS *will* operate in the new one-star mode during Relay and this backup is vitally important to proper Relay Antenna pointing in fault cases, in mid-September the spacecraft was operated in this mode to absolutely demonstrate its functionality.

15. Orbital Operations Software Development

The extensive new Phase 2 capabilities being developed to perform the Jupiter orbital mission with the low-gain antenna were discussed at the 1993 IAF Congress[2]. At that time the requirements and preliminary design were essentially complete and the detailed design and implementation were just beginning. Now, the implementation of all new flight and ground capabilities is nearly complete. The following is a brief progress report on how we did in the implementation relative to what we thought we could do two years ago.

Objectives of the Phase 2 modifications were to:

(1) increase science information density of the downlink using onboard editing and compression, coupled with a switch from time-division-multiplexed (TDM) to packetized telemetry,

(2) increase the number of downlink data rates and modes in order to utilize link capability efficiently,

(3) incorporate advanced coding techniques to increase telemetry return, and

(4) increase the DSN aperture and sensitivity for Galileo.

During these past two years, essentially all of these objectives have been achieved nearly as originally envisioned. Implementation in some areas, especially the Command and Data Subsystem (CDS), has been extraordinarily difficult due to the limited computational resources onboard the spacecraft.

Figure 12 presents a summary of the new Orbital Operations onboard spacecraft telemetry processing as implemented. All new telemetry processing capabilities including new software in the science instruments (and the supporting command system changes) have now been demonstrated using the science instrument breadboards or engineering models and the spacecraft system Testbed.

Figure 13 presents a summary of the associated Orbital Operations ground telemetry system, all elements of which have also been demonstrated in conjunction with system Testbed or other system level tests.

The salient Phase 2 accomplishments include:

- Eight (three of the eleven were not modified) science instruments delivered their new instrument FSW loads and have completed at least one functional system-level integration test on the spacecraft Testbed. Data from all but two instruments has been processed through the entire ground data system.
- Optical navigation (real-time and recorded/ playback) processing has been demonstrated on the Testbed, including the processing of those highly edited (oral y about 1 /400th of the original image is returned) Testbed output images on the ground system to reconstruct the optical navigation image.
- The CDS FSW was delivered (actually a phased delivery spread out over the last 13 months) and all functional elements demonstrated on the spacecraft system Testbed (editing, lossless compression, new record and playback processing, packet telemetry, and the new Reed-Solomon and convolutional encoding).

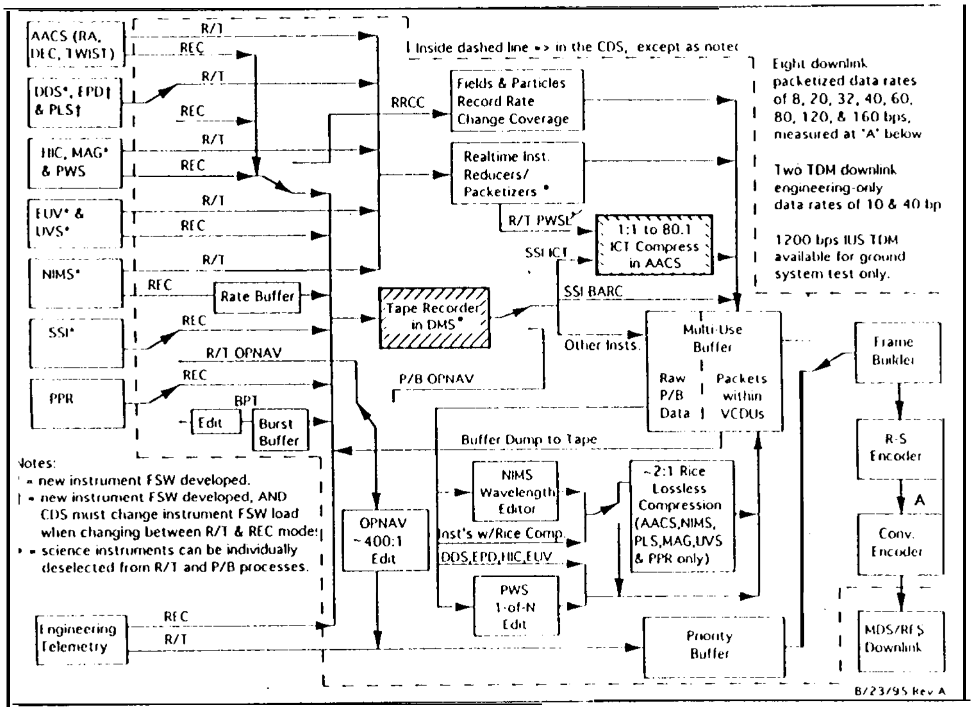


Figure 12. Orbital Operations (Phase 2) Spacecraft Onboard Telemetry Processing

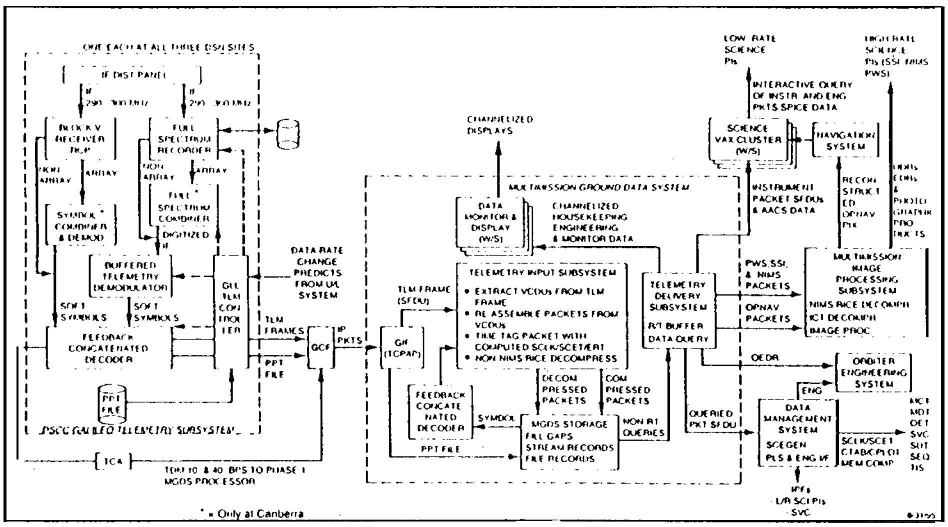


Figure 13. Orbital Operations (Phase 2) Telemetry Data Block Diagram

AACS	Attitude and Articulation Control Subsystem
BARC	Block Adaptive Rate Controlled image compression
BPT	Buffer Playback Table
CPLOT	Channel Plot
CLAB	Channel Tabulation
DDS	Dust Detector Subsystem
DMS	Data Memory Subsystem
DSN	Deep Space Network
EDR	Experimental Data Record
EPD	Energetic Particle Detector
ERT	Earth Received Time
EUV	Extreme Ultraviolet
GIF	GCF Interface
GCF	Ground Communications Facility
HIC	Heavy Ion Counter
ICT	Integer Cosine Transform
IF	Intermediate Frequency
IP	Internet Protocol
MAG	Magnetometer
MCT	Mission Control Team
MDS	Modulation Demodulation Subsystem
MDT	Mission Design Team
MGDS	Multimission Ground Data System
NIMS	Near Infrared Mapping Spectrometer
OEDR	Orbiter Engineering Data Record
OET	Orbiter Engineering Team
OPNAV	Optical Navigation
P/B	Playback
PLS	Plasma subsystem
PPR	Photo Polarimeter Radiometer
PPT	Post Pass Transfer file
PWS	Plasma Wave Subsystem
R/T	Real-time
RCP	Receiver Channel Processor
REC	Record
RFS	Radio Frequency Subsystem
RRCC	Record Rate Change Coverage
R-S	Reed-Solomon Code
SCEGEN	Spacecraft Event Time (SCE) Generation Software Module
SCET	Spacecraft Event Time
SCLK	Spacecraft Clock time
SDT	Science Data Team
SEQ	Sequence team
SFDU	Standard Formatted Data Unit
SPICE	Observation ancillary information
SSI	Solid State Imaging
SVC	Science Vax Cluster
TCA	Telemetry Channel Assembly
TCP/IP	Transmission Control Protocol/Internet Protocol
TDM	Time Division Multiplexing
TIS	Telemetry Input System
ULAR	Unprocessed Data Record
UVS	Ultra Violet Spectrometer
VCDU	Virtual Channel Data Unit
W/S	Workstation

because there are good reasons that suggest that this capability will not be needed.

- Currently, system level Testbed testing is proving a big challenge. Much expanded Phase 1 (arrival) testing along with late CDS FSW deliveries have compressed the planned Testbed testing schedule to approximately half of that originally planned. Additionally, problems discovered during initial testing of CDS RSW capabilities have led to the need for retests that are further packing an already full test schedule.

In summary, although the development of the new Orbital Operations capability has been an even greater challenge than expected, it now appears that a fully functioning flight and ground system capability will be ready in time to support the Galileo orbital tour as originally scheduled beginning in the spring of 1996.

16. Summary and Prospectus

Project Galileo has had an extremely successful and productive year. Telecommunications link capabilities required for Jupiter operations have been demonstrated. The new Flight Software required for Relay and JOI is installed and working perfectly. The Atmospheric Entry Probe was released in perfect health on a trajectory and attitude well within specifications. The 400-N main engine performed the Orbiter Deflection Maneuver (ODM) flawlessly. This first inflight bum of the engine demonstrated its performance for JOI. Propellant temperatures are now being held near constant to minimize the potential for any hazardous mingling of propellant vapors in the pressurization system because the oxidizer helium pressurant check valve is anomalously open. The "bulletproof" Relay/JOI Critical Engineering Sequence (CES) has been finalized and rigorously tested on the Spacecraft Testbed. Galileo is ready for arrival. The first of the Jupiter approach sequences is nowonboard. The remaining activities are uplinking the successive sequences, monitoring spacecraft sequence execution, performing the final Navigation/TCMs, and the most difficult remaining effort—the continuing preparation for contingencies that could affect Relay/JOI.

In addition to the extensive arrival preparations, the (Phase 2) Orbital Operations Flight Software has just been completed; and most all of the corresponding ground hardware and software is in hand, and five of the ten orbital tour encounter sequences have been completely designed.

Figure 14 illustrates the key spacecraft events in 1996 required to return the Atmospheric Probe data and then transition to the Jupiter Tour Orbital Operations. Note that the science observations recorded by the Orbiter on approach and during arrival are to be played back using the newly installed Phase 2 Flight Software beginning in mid-May 1996

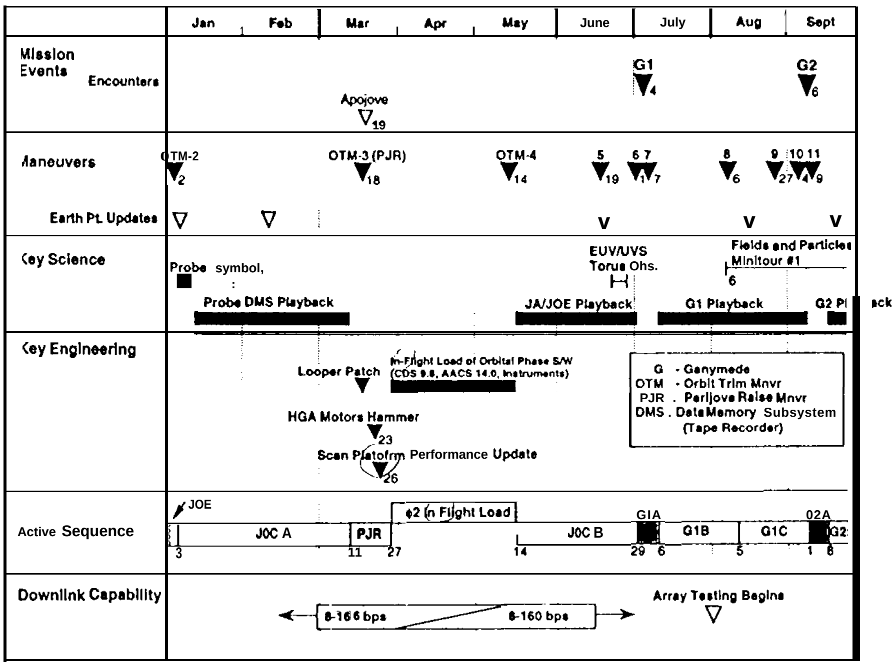


Figure 14. Galileo Timeline of Events (January 1996–September 1996)

and completing just in time for the first in-orbit satellite encounter—Ganymede-1—on July 4, 1996. (Some approach data may have to be carried over to the G1 playback, depending on downlink and image compression performance.) A high-level overview of the Galileo primary mission at Jupiter is given in Figure 15aand the Orbital Tour trajectory is illustrated in Figure 15b.

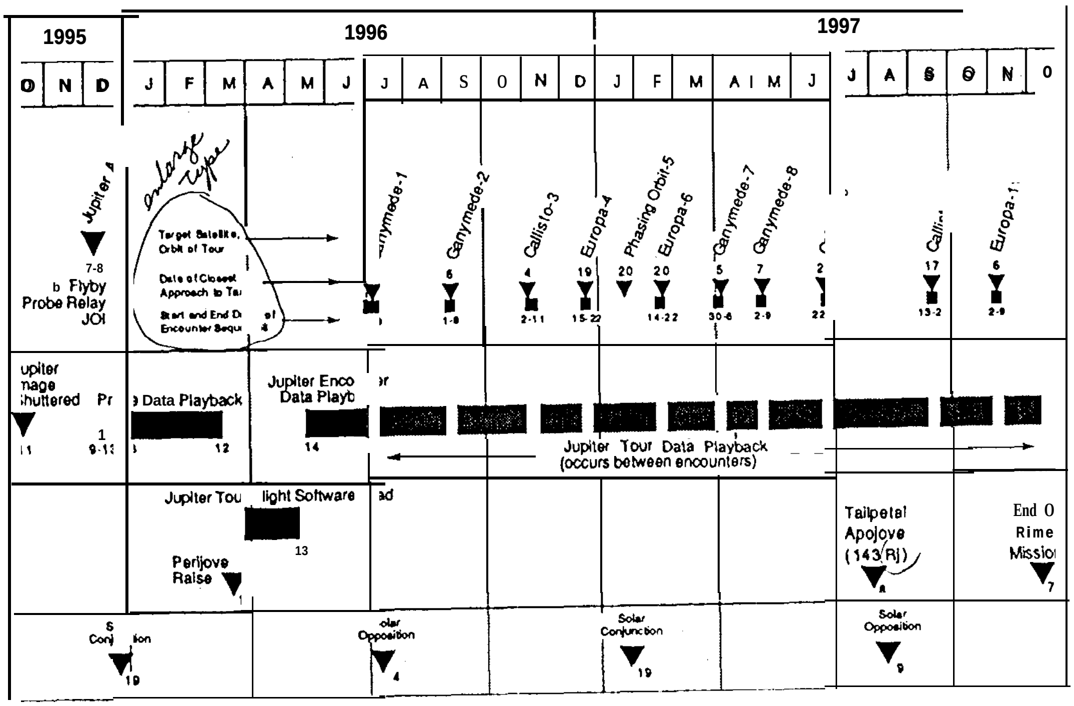
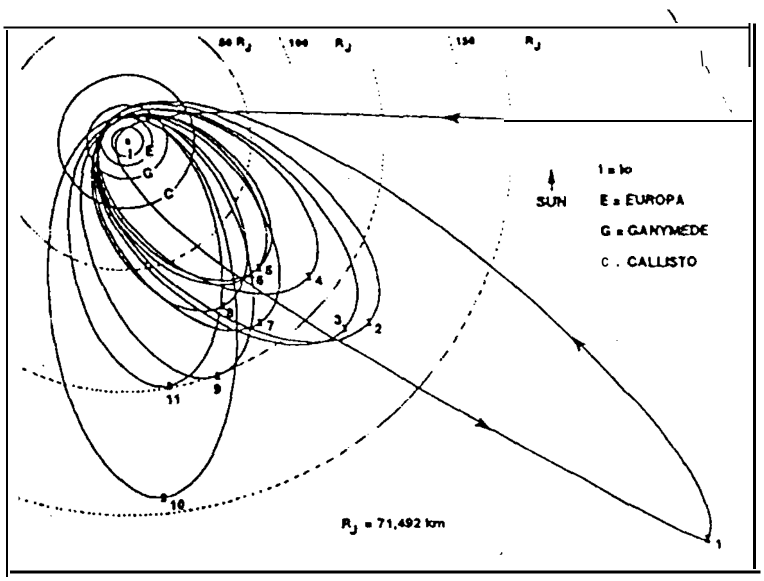


Figure 15a. Galileo Mission Overview (1995 to End of Mission)



17. Acknowledgment

Project Galileo represents the work of thousands of people. They all deserve acknowledgment for the many accomplishments of Galileo to date and for the tremendous promise it holds at Jupiter.

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